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An Integrated Cryogenic System for Spacecraft Power, Thrust, and Cooling

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Cryogenic hydrogen systems providing power can be used to advantage in spacecraft for mission durations ranging from one hour to over one hundred hours. By integration of the power, cooling, and attitude control requirements, the range of optimum application can be extended significantly. The maximum duration for optimum application depends upon vehicle attitude control requirements and the ability to utilize the large total impulses, at low thrust levels, contained in the working fluid discharged from an open-cycle expansion power unit. For example a 3-kw output oxygen-hydrogen expander operated at a BSPC of 1.12 lb per BHPHR will continuously provide 0.38 lb of thrust. If this can be used for attitude control purposes on a 50 per cent duty cycle, a system of this type will be optimum for durations up to 300 hr.

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An Integrated Cryogenic System for Spacecraft Power, Thrust, and Cooling

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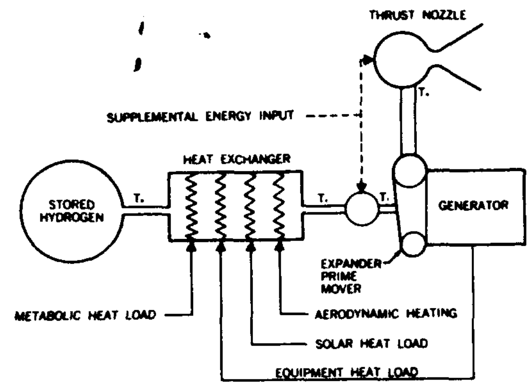

Fig. 1 Simplified system schematic

This paper is intended to demonstrate the capabilities of hydrogen in integrated systems providing cooling, secondary power, and thrust for certain classes of space vehicle. The applications where hydrogen can be used to advantage are characterized by high heat loads, moderate secondary power, and low thrust requirements. Manned reentry vehicles and recoverable boosters are examples of applications where substantial weight savings or improvements in vehicle mission capability can be effected.

The unique properties of hydrogen lead to high-performance, multiple-purpose systems which appear to offer potential advantage with respect to simplicity and reliability as well as weight. Since the announcement of the advanced booster programs utilizing hydrogen as the propellant, the objections to hydrogen, which were based upon the supply, storage and handling of a very low-temperature fluid, have largely been dispelled. In fact, use of hydrogen in secondary systems may now represent a reduction in the logistics problems since a single fluid, which will be available in large quantities at the launching complex, can perform the functions of a high-performance heat sink, working fluid, and propellant. Considerable weight savings are realized when the same hydrogen flow is used for all three functions. Attainment of maximum utilization of the hydrogen supply requires careful integration of all the vehicle systems.

Considerable interest has developed during the past two years concerning application of cryogenic hydrogen to secondary systems for space vehicles. References (1) through (6)[1] represent some of the published reports describing hydrogen secondary power systems which provide heat-sink capabilities for the environmental control system. Zwick (1) proposes use of a Rankine-cycle heat engine with sulfur as the working fluid, using a stoichiometric hydrogen-oxygen combustor as the energy source. Breaux (2), Hlavka (3), and Orsini (4) consider power conversion by means of multiple-stage turbines, using fuel-rich combustion products of hydrogen and oxygen as the working fluid in an open cycle. Howard (5) proposes use of a positive-displacement reciprocating engine as a hydrogen expander. Wood (6) presents the advantages of the positive-displacement expander at low power levels as well as a proposed performance referent based on an infinite pressure ratio.

References (7) through (13) give data concerning various properties of hydrogen.

DESCRIPTION OF SYSTEM

Fig.1 depicts the system which will be discussed. Hydrogen is stored in a supercritical state (typically 300 psi and 50 R) or as a liquid at lower pressures. The hydrogen is heated in a heat exchanger to provide environmental cooling capabilities. The heated hydrogen is expanded through a turbine supplying vehicle secondary power. The hydrogen discharged from the turbine is additionally expanded in a nozzle to provide thrust. The magnitude of the thrust, produced from hydrogen flows consistent with the secondary power and cooling requirements, is of the order needed for low-thrust attitude control.

It should be noted that with this system, useful power is derived from waste heat. An important factor in system design in determining the performance of hydrogen as a working fluid and propellant will be the maximum effective temperature of the heat source. In some applications requiring large amounts of secondary power or thrust, it may be desired to use an additional source of heat provided chemically by the combustion of hydrogen with an oxidant. Or, it may even be desired to increase the temperature of

[1] Underlined numbers in parentheses designate References at the end of the paper.

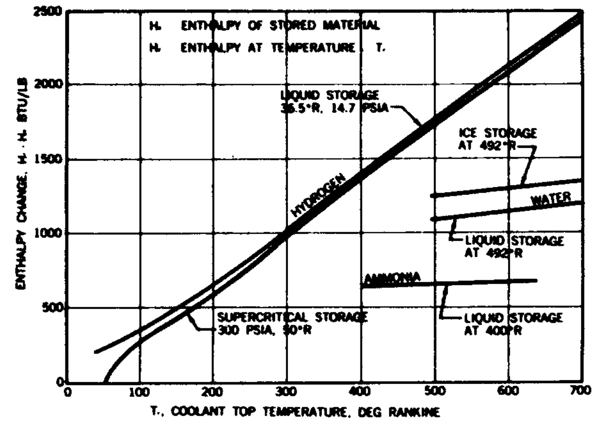

Fig. 2 Hydrogen, water and ammonia enthalpy change

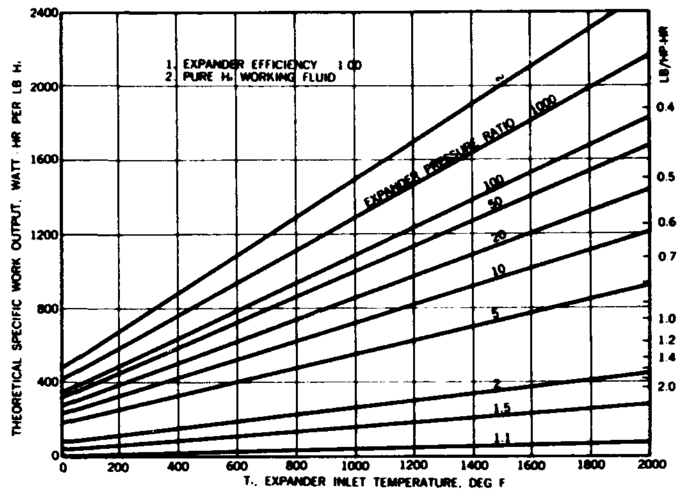

Fig. 3 Theoretical specific output for pure hydrogen cycle

the surfaces exposed to aerodynamic heating (that are also cooled by hydrogen) by selection of materials which minimize radiative heat losses or by aerodynamic design intended to maximize aerodynamic heating. This represents a reversal of conventional practice in the design of reentry bodies.

THERMODYNAMICS OF HYDROGEN

Liquid hydrogen occurs in two nuclear spin quantum states as orthohydrogen and parahydrogen (7). Normal hydrogen consists of 75 per cent orthohydrogen and 25 per cent parahydrogen. In the liquid state, orthohydrogen undergoes an exothermal transition to parahydrogen at rates high enough to cause considerable loss of liquid in storage. Consequently, the liquid-hydrogen plants have been designed to produce essentially pure parahydrogen.[2] Table 1 shows some of the properties of hydrogen:

TABLE 1 PROPERTIES OF HYDROGEN

	Para*	Ortho
Molecular Weight	2.016	2.016
Normal Boiling Point (14.7 psia)	36.5°R	36.8°R
Latent Heat of Vaporization	190.5 Btu per lb	191.8 Btu per lb
Critical Temperature	59.4°R	59.7°R
Critical Pressure	187.7 psia	191.7 psia
Enthalpy of Liquid at NBP (Relative to ideal parahydrogen gas at zero R)	-109 Btu per lb	-193 Btu per lb

Hydrogen, because of its low molecular weight, has exceptional properties as a working fluid for a turbine and as a propellant. Because of its high specific heat, hydrogen also has extremely high heat-sink capabilities as an expendable evaporant when heated from cryogenic storage temperatures. Fig.2 shows the enthalpy change as a function of temperature for hydrogen from two storage conditions: (a) as a liquid at the normal boiling point of 36.5 R and (b) as a supercritical-state fluid at 300 psi and 50 R. Little difference exists in the heat-sink capabilities between storage as a liquid and storage in the supercritical state; storage as a liquid will lead to lighter weight storage vessels because of the lower pressure, but will entail some problems with respect to zero-g liquid withdrawal which will be encountered in most space applications. Shown also in Fig.2, for purposes of comparison, are the equivalent data for water and ammonia, both of which are frequently used as expendable evaporants for environmental system heat sinks. The indicated superiority of hydrogen over ammonia and water is somewhat offset by the increased tankage weight required for hydrogen.

As a working fluid for expansion engines, hydrogen has behavior closely approximated by the perfect gas law, since the operating conditions will be far removed from the critical state. Thus, for pure hydrogen, the work output is given by the following expression:

$$\text{Specific work output} = \frac{k}{k-1}\,\frac{R}{M_w}\,T_t\,\eta_t\left[1-\left(\frac{P_n}{P_t}\right)^{\frac{k-1}{k}}\right] \qquad (1)$$

where

- k = specific heat ratio
- R = gas constant = 1544 ft-lb per lb mole deg R
- M_w = molecular weight of working fluid = 2.016 for pure hydrogen
- T_t = expander inlet total temperature, deg R

[2] 99.79 per cent parahydrogen and 0.21 per cent orthohydrogen.

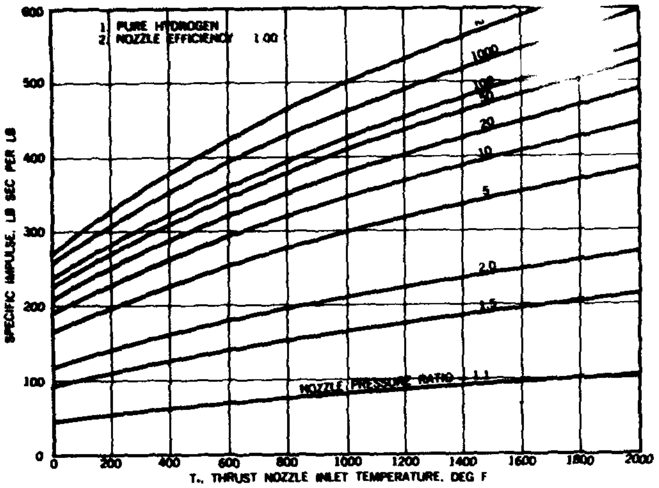

Fig. 4 Thrust nozzle specific impulse, hydrogen as a propellant

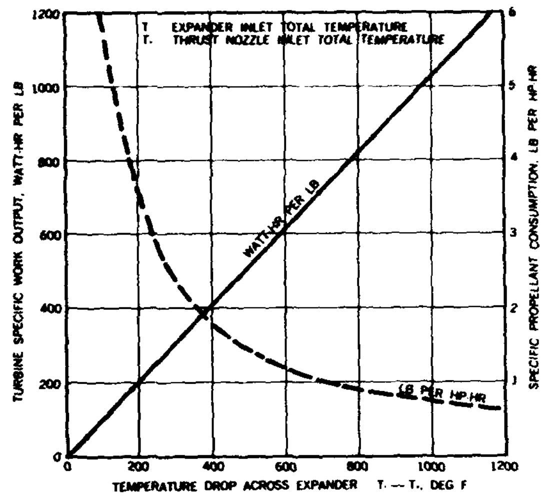

Fig. 5 Temperature drop across expander

η_t = expander adiabatic efficiency
P_t = expander inlet pressure
P_n = expander discharge pressure

Fig.3 shows the performance of hydrogen as a working fluid in a turbine as a function of pressure ratio (P_t/P_n) and turbine-inlet temperature (T_t °R) for an expander efficiency of 100 per cent. The performance shown is much higher than that of any monopropellant which is suitable for use with secondary power units. Hydrazine has the highest performance of the APU monopropellants and gives specific outputs of the order of 150 to 200 watt-hr per lb at a turbine-inlet temperature of 1800F and very low discharge pressures.

The specific impulse for pure hydrogen as a propellant is given by the following equation:

$$\text{Specific impulse} = \left\{ \frac{2k}{k-1} \frac{R}{g M_w} T_n \eta_n \left[1 - \left(\frac{P_a}{P_n} \right)^{\frac{k-1}{k}} \right] \right\}^{\frac{1}{2}} \quad (2)$$

where

g = gravitational acceleration = 32.2 ft per sec^2
T_n = thrust-nozzle inlet total temperature, deg R
P_n = thrust-nozzle inlet pressure
P_a = thrust-nozzle discharge pressure
η_n = thrust-nozzle efficiency

Fig.4 shows the theoretical specific impulse of hydrogen as a function of nozzle inlet temperature (T_n °R) and nozzle pressure ratio (P_n/P_a). High specific impulse is given, even at low temperatures and pressure ratios.

The temperature drop across the prime mover, which relates the thrust-nozzle inlet temperature

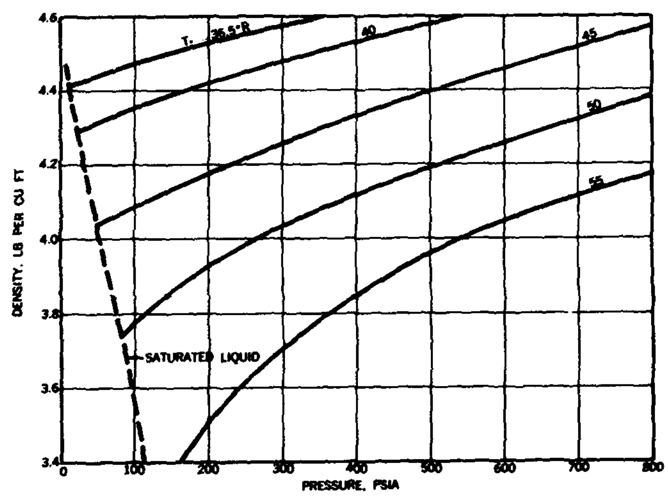

Fig. 6 Hydrogen storage density

and the expander inlet temperature, is given in Fig.5 as a function of expander specific output. It may be desired to increase the energy level of the gas for propulsion by additional combustion with oxygen.

Fig.6 shows the density of hydrogen as a function of storage temperature and pressure. The low density of hydrogen and the extremely low temperatures involved in its storage are the main disadvantages in application of hydrogen.

Fig.7 gives the net cooling capacity as a function of expander power output. The net cooling capacity must be supplied to the system in order for the power unit to supply the specified output under the given conditions. In the useful temperature range, the expander inlet temperature has little influence on the net cooling capacity because, at the higher inlet temperatures, the

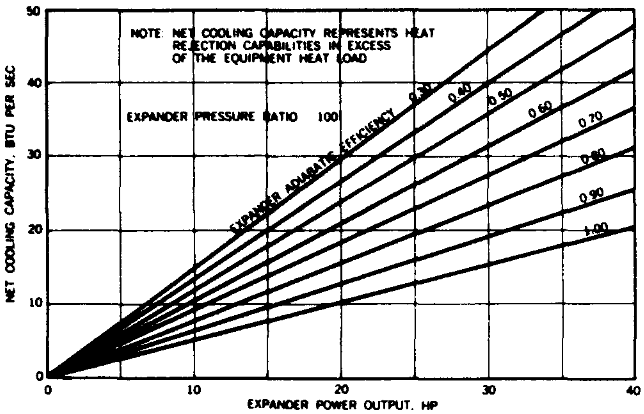

Fig. 7 Net cooling capacity versus turbine output

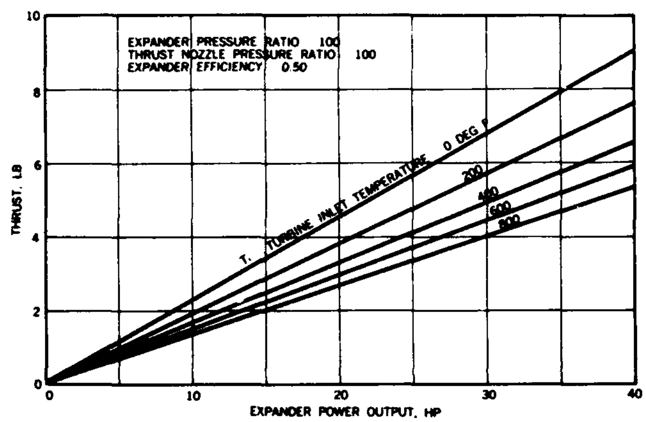

Fig. 8 Thrust versus expander output

expander specific-power output offsets the increased enthalpy of the hydrogen.

Fig.8 shows the available thrust as a function of expander power output for specified expander and thrust-nozzle pressure ratios (both = 100). Higher expander inlet temperature is accompanied by reduced thrust, since the increased specific impulse (proportional to $T_t^{1/2}$) is more than offset by the lower hydrogen flow resulting from the increased expander specific output (proportional to T_t).

PROPELLANT TANKS

As is generally realized, tankage size and weight are of major importance in the design of oxygen-hydrogen systems. Both of these fluids are best stored at cryogenic temperatures in order to obtain maximum density. Minimization of tankage weight thus involves careful thermal design to avoid propellant loss by vaporization or venting. The hydrogen tank requires the most

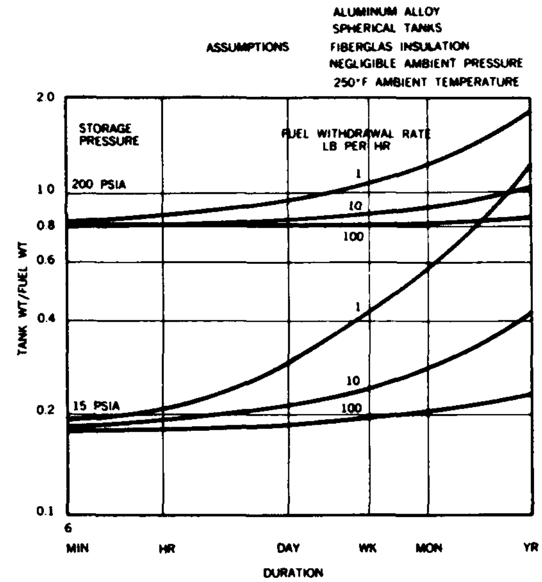

Fig. 9 Hydrogen tank weight

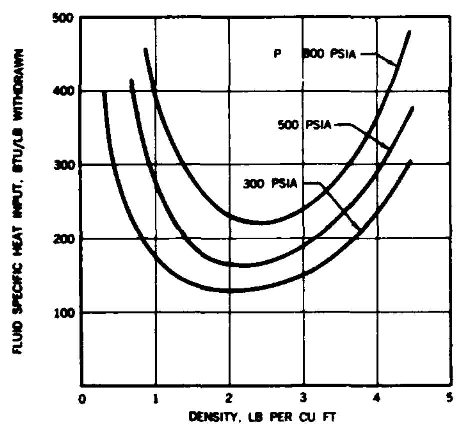

Fig. 10 Fluid heat input requirements for constant pressure operation supercritical hydrogen storage

consideration in this respect because of its relatively large volume.

The space-vehicle environment is particularly favorable for cryogenic storage in one major respect. The tank ambient is a practically pure vacuum; the only heat transfer to the fuel (aside from conduction through a structural support) is by radiation from compartment walls and external equipment. This eases the insulation problem considerably.

The zero gravity condition associated with space flight affects the fuel-tank design in two respects: (a) If the fuels are stored as liquids, positive design techniques must be employed to insure single-phase withdrawal. (b) It necessitates pressurized transfer. Tankage design

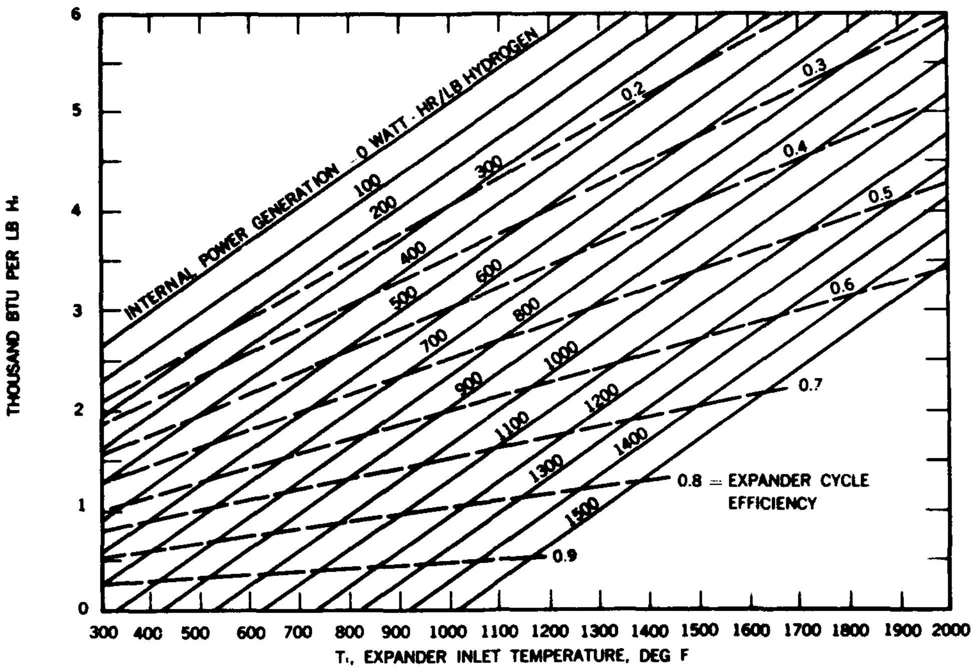

Fig. 11 Supplemental heat requirements for pure hydrogen cycle

studies and tests have indicated that two general methods of fuel storage can be considered most suitable: There are (a) low-pressure storage of the fluids as liquid using expulsion bladders and (b) supercritical storage.

Fig.9 is an estimate of hydrogen tankage weight for two storage pressures as a function of time and withdrawal rate. Supercritical storage permits agravic operation and elimination of hydrogen pumps. However, as shown in Fig.10, substantial energy inputs are required to maintain supercritical conditions. Since the peak heat loads and maximum power demands do not necessarily coincide, some form of supplemental heat input or cooling may be required during portions of the mission.

SYSTEM HEAT INPUT

The total heat input to the hydrogen working fluid is derived from three sources: The internal heat load, represented by the equipment and metabolic loads; the external heat load, represented by aerodynamic heating and solar radiation loads; and supplemental heat input, represented, for example, by combustion of hydrogen with stored oxygen (to bring the working fluid to the optimum temperature range for maximum utilization of the hydrogen). The equipment heat load is equal to the energy extraction from the working fluid in the prime mover. Thus, the sum of the energy inputs from the other sources must be equal to the increase in enthalpy of the material (from storage conditions) leaving the system. This additional heat input is depicted in Fig.11 for the pure-hydrogen cycle as a function of cycle efficiency where all of the power generated by the expander is absorbed in heating of the hydrogen. Cycle efficiency is defined as the product of the prime-mover adiabatic efficiency and the isentropic expansion factor, equation (1). For example, if the internal power generation represents an output of 500 watt-hours per lb (1704 Btu/lb) with an expander inlet temperature of 300 F, approximately 900 Btu must be supplied to the system per pound of hydrogen. It will be found that in many instances, the additional energy provided by combustion with oxygen will be required for at least a portion of the mission duty cycle.

Fig.12 shows the oxygen-hydrogen weight ratio required to attain a given expander inlet temperature as a function of combustor inlet temperature (coolant top temperature). Simple expansion heat engines will be limited to relatively low inlet temperatures which means that the cycle will usually be operated fuel rich, at oxygen-hydrogen ratios less than one.

Fig.13 gives the correction factors for oxygen-hydrogen combustion products to be applied to the specific impulses and to the specific work outputs given by Figs.3 and 4 for pure hydrogen. For example, if an oxygen-hydrogen ratio of 1.0 is required to reach an inlet temperature of

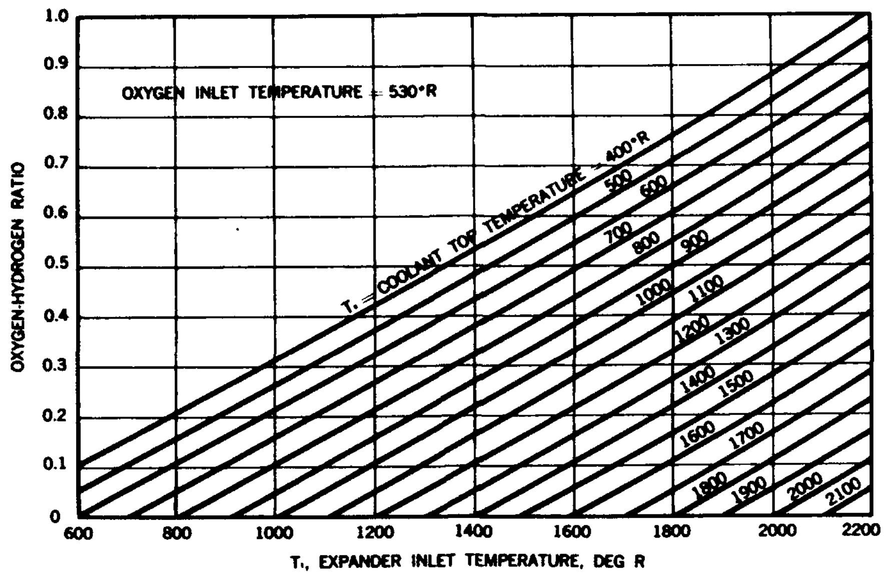

Fig. 12 Oxygen requirements for bipropellant cycle

TABLE 2
SPECIFIC PROPELLANT CONSUMPTION REQUIRED FOR ENERGY BALANCE

COOLANT TOP TEMP, DEG F	SPC (LB PER BHP HR) TO GIVE ENERGY BALANCE					
	O/H = 0	O/H = 1	O/H = 2	O/H = 4	O/H = 6	O/H = 7.94
100	1.43	2.61	3.58	5.13	6.30	7.15
200	1.20	2.19	3.03	4.38	5.37	6.13
300	1.03	1.89	2.62	3.79	4.67	5.37
500	0.81	1.49	2.06	3.00	3.72	4.27

1800 F, a correction factor of 0.53 is applied to the specific work output for pure hydrogen, which is 2310 watt-hours per lb (0.323 lb per hphr) at an infinite pressure ratio, reducing it to 1230 watt-hours per lb (0.607 lb per hphr). Similarly, the correction factor for the specific impulse is 0.727, which reduces the specific impulse from 635 sec to 461 sec, both at an infinite pressure ratio and a thrust-nozzle efficiency of 1.0. Cycle efficiencies, represented by the product of the adiabatic efficiency and the isentropic expansion factor, must be applied to the theoretical specific work and specific impulse to indicate realizable performance. The correction factors given in Fig.13 are applicable to gas temperatures in the 200-2500 F range.

Fig.14 shows the variation in the heat-sink capacity of the propellants as a function of the oxygen-hydrogen ratio and the coolant top temperature. Table 2 illustrates the limitation on prime-mover specific propellant consumption, if the propellant flow is to be used as the heat sink for the environmental control system. Fig. 14 and Table 2 are based upon an energy balance between heat transferred to the propellant flow and power-unit output, as follows:

$$\text{SPC for energy balance} = \frac{Q}{P}\,\frac{2545\,[(O/H) + 1]}{(H'_c - H'_o) + (O/H)(H''_c - H''_o)} \qquad (3)$$

where

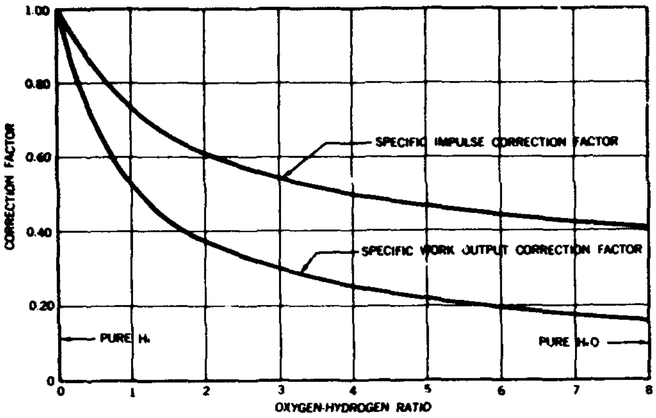

Fig. 13 Correction factors for combustion products

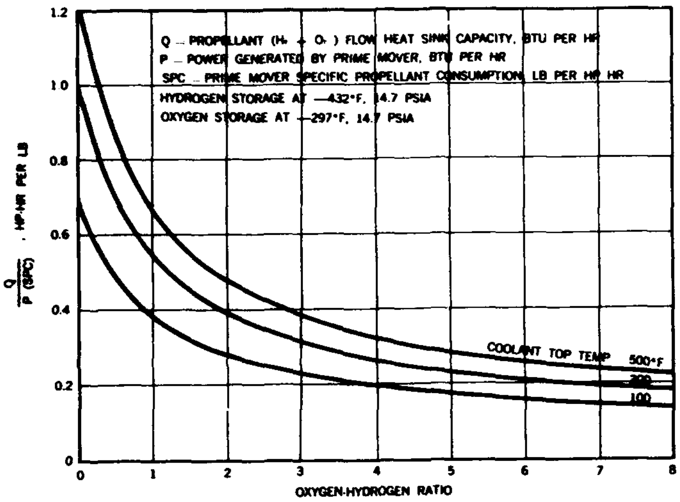

Fig. 14 Propellant cooling capacity

Q/P = fraction of expander shaft-power output absorbed as heat in propellant flow

O/H = oxygen-hydrogen ratio

$H'_c - H'_o$ = enthalpy change in heating hydrogen to coolant top temperature

$H''_c - H''_o$ = enthalpy change in heating oxygen to coolant top temperature

Equation (3) is based upon an energy balance and does not indicate attainable prime mover performance, which is given by equation (1).

HYDROGEN POWER CYCLES

Hydrogen can be utilized in a number of different cycles for the generation of power. The principal methods of interest in secondary-power applications can be classified in three categories, as follows:

1 Pure hydrogen systems.

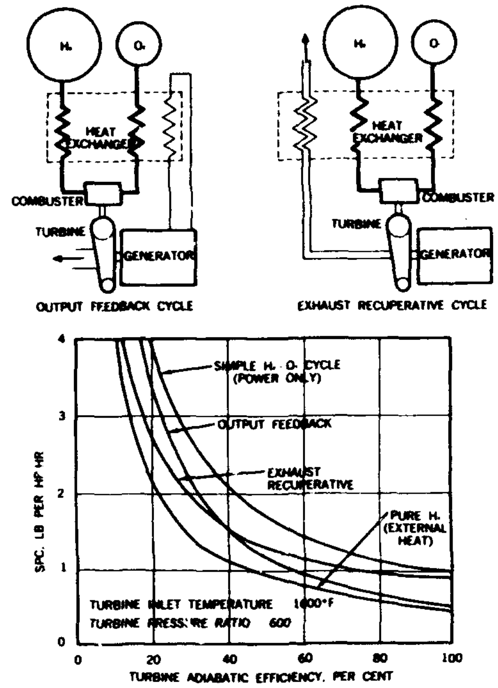

Fig. 15 Performance of four hydrogen systems

2 Stoichiometric oxygen-hydrogen bipropellant systems:
   - Open-cycle heat engines:
     - Otto and diesel cycles.
   - Closed-cycle heat engines:
     - Rankine and stirling cycles.
   - Open-Cycle fuel cells:
     - Direct and indirect types.
   - Compound conversion cycles:
     - Thermionic (or MHD) and thermoelectric.

3 Fuel-rich oxygen-hydrogen bipropellant systems:
   - Exhaust recuperative.
   - Output feedback regenerative.
   - Interstage reheat.
   - Simple open cycles.

Pure hydrogen systems can be used where relatively large amounts of waste heat are available for heating of the hydrogen. The stoichiometric bipropellant systems have high performance in terms of high specific outputs with respect to both weight and volume, but provide comparatively little cooling capacity. The fuel-rich bipropellant systems provide high performance in terms of secondary power, cooling and thrust.

The performance of four different types of hydrogen power cycle is shown in Fig.15. In the simple oxygen-hydrogen system, there is a combustor in which oxygen is added to the hydrogen and burned to increase the temperature of the working fluid. In the pure-hydrogen system, the oxygen

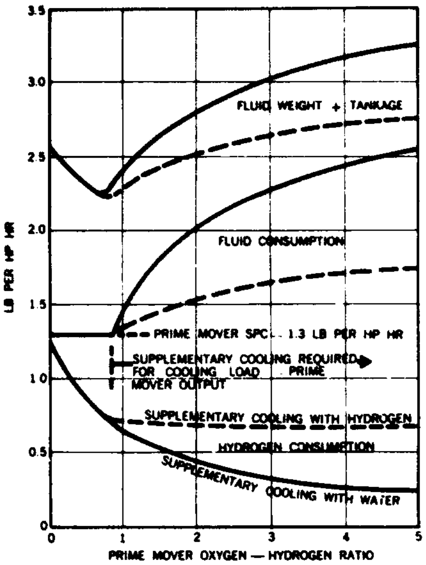

Fig. 16 Optimum oxygen-hydrogen ratio for SPC = 1.3 lb per hp hr considering heat-sink requirements

has been eliminated completely, thus affording the opportunity for minimum propellant weight. Of course, heat must be provided from an external source, as previously discussed. There are many ways to improve performance of the simple oxygen-hydrogen cycle; two are shown in Fig.15, the exhaust recuperative cycle and the output feedback cycle. In both instances, waste heat is utilized for propellant heating. In the first instance, the prime-mover exhaust acts as the heat source; in the second case, the equipment heat load is the heat source.

Stable combustion of oxygen and hydrogen can be accomplished over a wide range of conditions with respect to pressure and mixture ratio. Stoichiometric combustion of hydrogen and oxygen is accompanied by a high heat of reaction, with the flame temperature limited by the thermal dissociation of water to 4800 F at 14.7 psia (6500 F at 1000 psia). A number of different types of conversion device can be designed to accommodate near-stoichiometric oxygen-hydrogen ratios, as previously mentioned. Thermodynamic calculations, such as those shown in Fig.14, will show that operation at stoichiometric ratios produces useful cooling representing something less than 25 per cent of the power output.

At a specific propellant consumption of 1.3 lb per bhphr, an oxidizer-fuel ratio of 0.85 will provide sufficient cooling capacity, in the propellant flows, to absorb the power output as a heat load. At oxygen-hydrogen ratios in excess of 0.85, supplemental cooling will be required. This is illustrated in Fig.16 for cooling with hydrogen and with water as expendable evaporants, assuming the following allowances for tankage as a fraction of fluid weight:

Liquid-hydrogen tankage	1.00
Liquid-oxygen tankage	0.33
Water tankage	0.11

Fig.16 shows a minimum total weight (propellants + evaporants + tankage) at an oxygen-hydrogen ratio near 0.85 (for a prime mover SPC = 1.3 lb per bhphr) where no supplemental cooling is required. However, it will be noted that the hydrogen flow decreases with increasing oxygen-hydrogen ratio which will reduce the storage volume required. Consequently, there appears to be a trade-off between weight and volume, and system selection may be based upon volume penalty factors, which will vary with the application.

ENVIRONMENTAL SYSTEM HEAT REJECTION

The environmental control system removes waste heat (sensible and latent) ultimately rejecting the energy to a heat sink of one of the following four types:

1 Low-temperature radiator - transfer by conduction.

2 Low-temperature radiator - transfer by cooling loop.

3 High-temperature radiator - transfer and elevation of temperature level by vapor-cycle heat pump, thermoelectric converter or other energy conversion device.

4 Expendable evaporants or propellants.

Early space vehicles have utilized direct radiation from the skin to dissipate the heat which was transferred by conduction from the equipment. By selection of surface coatings which demonstrate high emissivity to the long wave length thermal radiation emitted by the skin and which simultaneously show high reflectivity to the comparatively short wave length solar radiation, it has been possible to achieve the desired internal temperatures with purely passive systems. Some of these vehicles have incorporated simple bimetallic strip-actuated shutters to vary the emissivity of the exposed surface as a temperature control. Clearly, the simple, direct-conduction, low-temperature radiator will be limited in application to vehicles with small payloads, low heat loads, and relatively simple equipment with broad environmental control requirements.

Use of low-temperature radiators with intermediate coolant loops will be desired for many applications because of installation flexibility and ease of control. Fig.17 shows typical per-

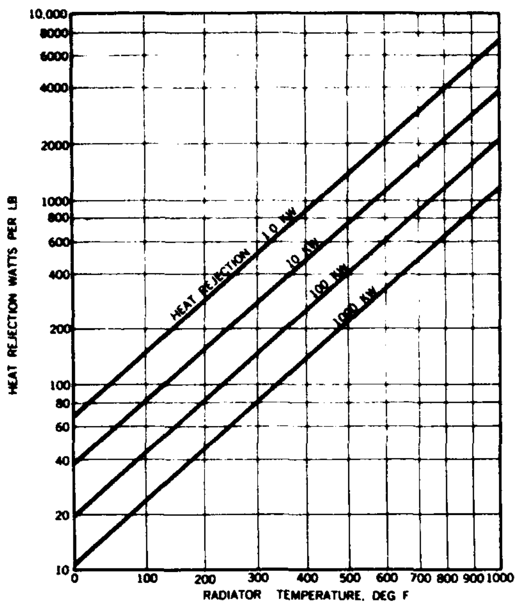

Fig. 17 Radiator heat rejection for radiation to free space

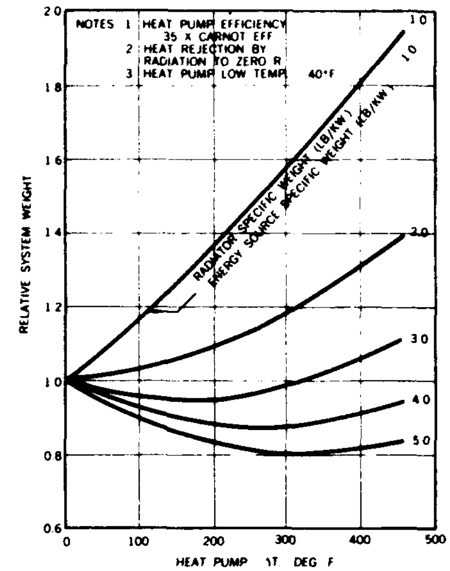

Fig. 18 Relative system weight for high temperature radiator using a heat pump

TABLE 3

PROPERTIES OF VARIOUS EVAPORATIVE HEAT SINKS AND WORKING FLUIDS

	Density at Boiling Point lb/cu ft	Normal Boiling Point deg F	Critical Point psia	Critical Point deg F	Freezing Point deg F	Latent Heat Btu/lb	Thermal Capacity Btu/lb	Maximum Available Energy in Expansion Btu/lb
Water	59.7	212	3206.2	705.4	32	970	1148 (4)	346 (2)
Ammonia	42.6	-28	1657	271.4	-108	590	745 (3)	418 (2)
Nitrogen	50.5	-320	492.3	-232.9	-346	85.5	240 (3)	191 (2)
Oxygen	71.3	-297	730.6	-181.7	-361	91.6	224 (3)	172 (2)
Hydrogen	4.4	-423	187.7	-400.3	-434	191	2465 (3)	2580 (2)
Helium	8.3	-452	33.2	-450.2	-458	9.4	954 (3)	949 (2)
Ethylene Oxide (1)	55.1	51	1043	384	-171	249	--	1470 (1)
Hydrazine (1)	58.5	236	2130	717	34	550	--	1490 (1)

(1) Monopropellants

(2) For infinite pressure ratio, expander efficiency = 1.0, and an inlet temp = 300F

(3) Heat from boiling point temp to 300F at 300 psia

(4) Boil at 32F, superheat to 300F

formance for finned-tube radiator surfaces, where the intermediate heat-transfer fluid is pumped through the tubes. The increase in radiator specific weight with heat dissipation results from the increased tube-wall thickness required for protection against micrometeorite penetration. That is, in this case, the tube-wall thickness is sized for a 95 per cent probability of survival of one year and increasing surface area represents an increased chance of impact with larger particles.

The desirability of using a heat pump with a high-temperature radiator will depend upon the ratio of the radiator specific weight to the electrical energy source specific weight, as shown in Fig.18. Weight reductions are obtained for values of this ratio in excess of approximately 3, which requires electrical energy sources with specific weights of less than 5 lb per kw_e for 1 kw heat dissipation, 9 lb per kw_e for 10 kw, 17 lb per kw_e for 100 kw, or 30 lb per kw_e for 1000 kw, assuming an environmental system sink temperature of 0 F. Since these specific weights are low, relative to those currently obtainable, the near-term prospects for heat pumps as a means of reducing weight do not appear to be favorable, except possibly for components requiring very low rejection temperatures or for

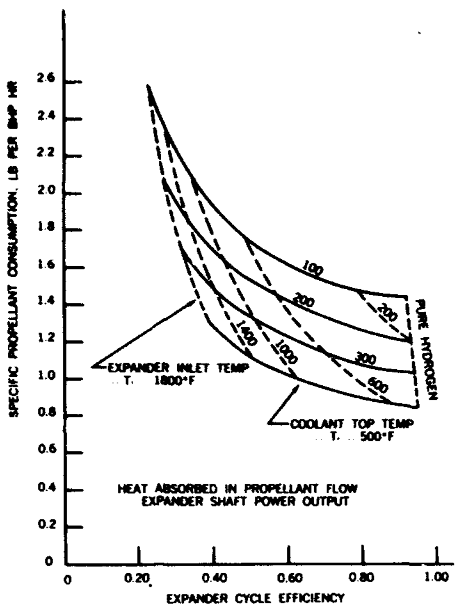

Fig. 19 Integration of power and cooling for Q = P

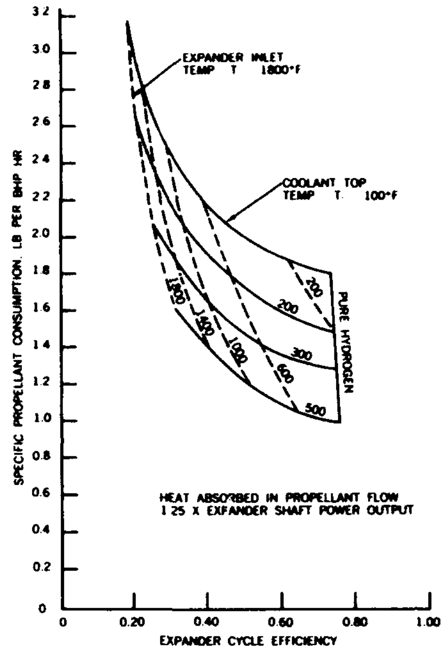

Fig. 20 Integration of power and cooling for Q = 1.25P

the case where the temperature of the radiative heat sink is higher than the desired source temperature.

Expendable evaporants will be at a disadvantage with respect to weight as compared with radiators for long-duration space missions where radiation to free space is possible. For example, assuming a radiator specific weight of 25 lb per kw_t, and neglecting tankage, radiators will be superior to evaporants for durations in excess of 7.3 hr for water and in excess of 11.9 hr for hydrogen. The expendable evaporant may be utilized as a working fluid in a power cycle before being dumped overboard. Table 3 contains a comparison of several fluids for use both as a heat sink and as a working fluid. The advantages of hydrogen over other fluids, in both capacities, is evident.

SYSTEM INTEGRATION

System integration is a critical factor in the design of multipurpose systems where the output requirements can vary more or less independently, but within limits which can be specified for a particular mission. Inter-relationships exist between certain of the steady-state requirements, as between the internal heat load and the power generation, for example. However, the possible independence of the output requirements, particularly during transient conditions, means that provision will usually be required for supplemental energy sources and heat sinks. The concept of an integrated multipurpose hydrogen system is attractive in that it utilizes a high-performance fluid to the maximum extent for a variety of functions. Obviously, optimum propellant utilization for all functions will probably not be possible in most applications.

The heart of system integration involves the energy balance between the heat input to the propellant flow and the internal power generation. Expander cycle efficiency and coolant top temperature are important parameters in determining attainable specific propellant consumptions, as shown in Fig.19 for the case where the expander power output is absorbed in heating of the propellant flow. Expander cycle efficiency is defined as the product of the prime-mover adiabatic efficiency and the perfect-gas isentropic-expansion factor, as follows:

$$\text{Cycle Efficiency} = \eta_t \left[1 - \left(\frac{P_n}{P_t} \right)^{\frac{k-1}{k}} \right] \quad (4)$$

where

η_t = expander adiabatic efficiency
P_t = expander inlet pressure
P_n = expander discharge pressure

Assuming efficiencies in the 0.50-0.60 range to be realizable at the vacuum discharge conditions of space, specific propellant consumptions of the order of 1.40-1.53 lb per bhphr are indicated for a coolant top temperature of 200 F. It should be noted that under these conditions

the prime mover will be operating with inlet temperatures in the 600-800 F range.

In most space vehicles, the total heat load will be greater than the equipment heat load represented by internal power generation. The additional heat load is given by crew metabolism, solar radiation, and aerodynamic heating. Fig.20 illustrates the effect of a heat load 125 per cent of the expander shaft power output on system performance. For example, using the previously assumed expander cycle efficiency (0.50-0.60) and coolant top temperature (200 F), specific fuel consumptions for power generation now fall in the 1.61-1.73 lb per bhphr range, showing a penalty of approximately 0.20 lb per bhphr. This penalty corresponds to a "specific evaporant consumption" of 0.80 lb per hphr as compared with 2.36 lb per hphr for water and 1.21 lb per hphr for hydrogen as expendable evaporants providing cooling alone.

For long-duration applications, it may be preferred to provide a part of the cooling by radiation to free space to minimize propellant weight. Reduced specific propellant consumptions can be obtained for reduced cooling capacity, as shown in Fig.21 for the case where the heat input to the propellant flow represents 75 per cent of the expander shaft power output. The specific propellant consumption will now fall in the 1.21-1.33 lb per bhphr range for the coolant top temperature of 200 F and expander cycle efficiency of 0.50-0.60. The expander inlet te~~eratures are higher for this condition of operation, falling in the 900-1200 F range, but are still below the maximum design values for long-life, reliable turboexpanders.

For minimum specific propellant consumption at a given expander cycle efficiency, expander inlet temperature will be a limiting factor, as illustrated in Fig.22 for a cycle efficiency of 50 per cent and a maximum expander inlet temperature of 1800 F. In this case, a specific propellant consumption of 1.12 lb per bhphr is shown for the previously assumed coolant top temperature of 200 F with Q/P = 0.54. A radiator or other supplemental heat sink will be required to dissipate the remainder of the heat load under this condition of operation.

From the previous analysis, the following summation can be made for a coolant top temperature of 200 F:

Q/P	SPC
1.25	1.73 lb/bhp hr
1.00	1.53
0.75	1.33
0.54	1.12 T_t = 1800 F

Assuming radiator specific weights of the order of 25 lb per kw_t, it appears that the minimum specific propellant consumption will be desired for mission durations in excess of approximately 19 hr. For missions of less than 19 hr length, it may be expedient to provide all of the cooling required in the propellant flow since the resultant increase in specific propellant consumption will be more than offset by weight reduction in the environmental control system.

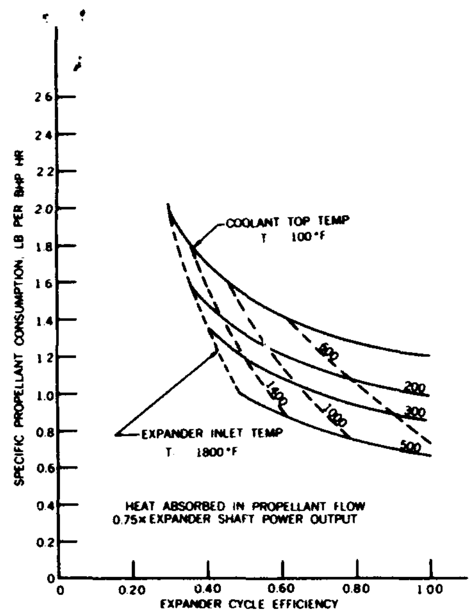

Fig. 21 Integration of power and cooling for Q = 0, 75P

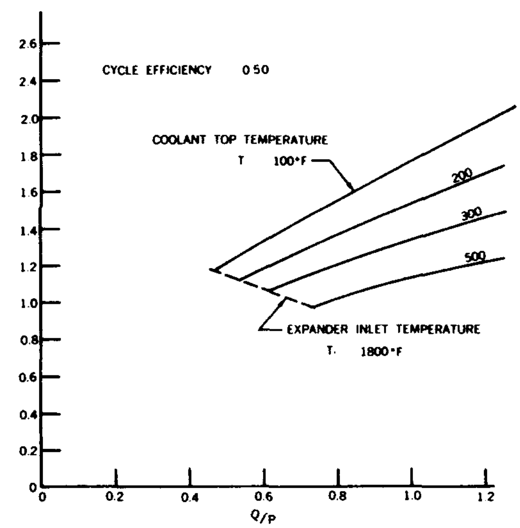

Fig. 22 Integration of power and cooling for cycle efficiency = 0. 50

POWER-SYSTEM COMPARISON

Fig.23 compares various types of power system in terms of specific work output as a function

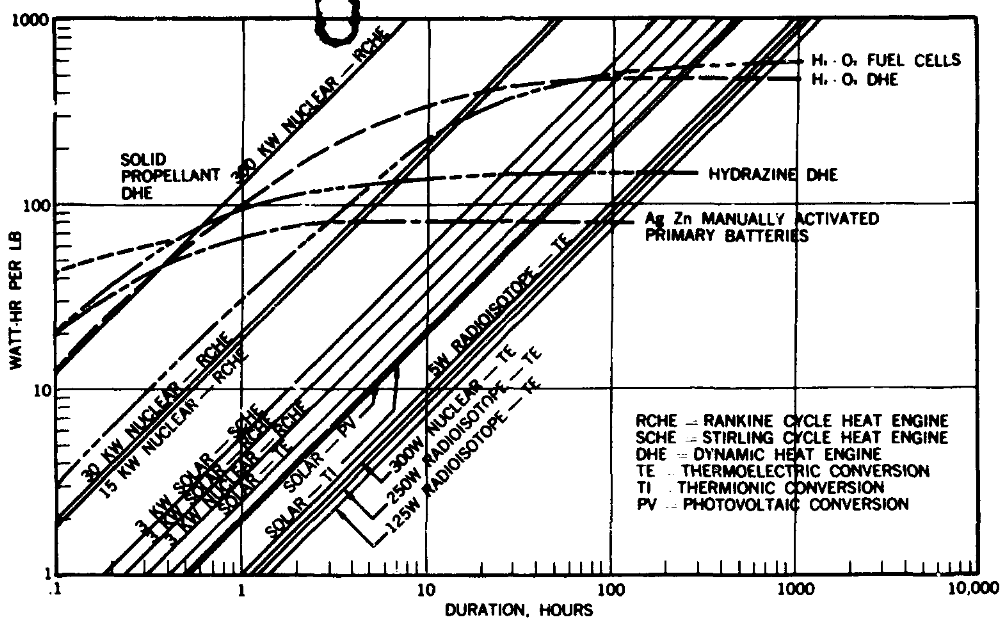

Fig. 23 Comparison of space-vehicle power systems

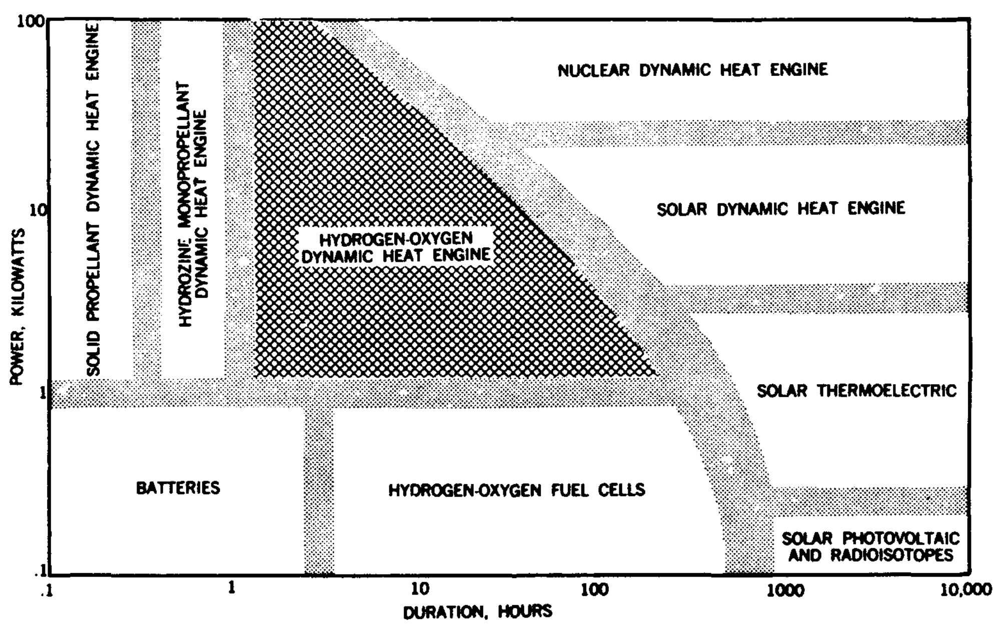

Fig. 24 Space-vehicle power systems--estimated optimum application

of duration. Fig.23 is based upon the requirements for power generation only; the vehicle mission and the environmental-control and attitude-control requirements may affect system selection. For example, space vehicles which reenter the atmosphere will require some type of chemical APU and heat sink (in addition to a solar or nuclear power unit used in space flight) to provide power and cooling during the boost and reentry phases of operation.

From the data presented in Fig.23, it is possible to establish, within rather broad limits, optimum power systems. Fig.24 is an estimate of optimum power systems as a function of power output and duration. It can be seen that for durations less than 1 hr, battery, solid-propellant, and monopropellant systems will be used to advantage. For extremely long durations, solar and nuclear systems will be optimum. Oxygen-hydrogen secondary systems appear to be the optimum means

TABLE 4
TYPICAL REQUIREMENTS -
SPACECRAFT POWER, ENVIRONMENTAL CONTROL, AND ATTITUDE CONTROL

MISSION: Earth Orbit

OCCUPANTS: 3 Men

SECONDARY POWER:

Boost, Injection, Deployment and Orientation	1 hr	3 kw
Orbital Flight	Variable	3 kw
Emergency/Standby	4 hr	3 kw
Separation, Maneuvering, and Re-Entry	1 hr	3 kw
Post-Landing	12 hr	0.2 kw

HEAT LOADS:

Occupants: 1800 Btu per hr

Solar Radiation: Zero

Re-Entry: 100,000 Btu

Equipment: 3 kw + APU Heat Generation

CAPSULE ENVIRONMENT:

Occupied Area: 80°F maximum

Equipment Area: 220°F maximum

Pressure: Sea Level to 8 psia

ATTITUDE CONTROL:

Low Thrust, Long Duration: 0.2 lb average (continuous)

High Thrust, Short Duration: 1000 lb sec

of power generation for the very important class of space mission ranging from 1 hr to over 100 hr. In the section following, three types of power system will be compared, with the objective of determining the effect that mission, attitude control, and environmental control requirements have on power-system selection in integrated spacecraft systems.

INTEGRATED SYSTEM COMPARISON

For the purposes of comparing three different integrated systems providing power, environmental control and thrust, the design requirements listed in Table 4 were assumed. The three systems to be compared are (a) oxygen-hydrogen bipropellant expander system, (b) oxygen-hydrogen fuel-cell system, and (c) solar power-unit system.

The basic environmental control system used in all three systems is essentially the same. It consists of circulating fans, heat exchangers, filters, oxygen and nitrogen tanks and controls; it provides for internal temperature control and atmospheric control and makeup. The atmosphere control subsystem incorporates means for carbon dioxide and water-vapor removal, odor control, sterilization and prevention of trace contaminant buildup. The atmospheric makeup subsystem has a cryogenic oxygen supply and controls to maintain the proper partial pressure of oxygen in the atmosphere. Also included is a cryogenic nitrogen supply for makeup of leakage.

To maintain the vehicle on its orbital course and correct for minor perturbations, a continuous control thrust of the order of 0.2 lb is assumed to be required from the attitude control system. Also, to correct for major disturbances that may occur during separation and for orientation during reentry, a high thrust is required, assumed to be 50 lb for 20 sec duration.

The equipment required for emergency or standby operation may represent an important factor in providing the high reliability required of manned vehicle systems. Selective component or system redundancy will be required, with particular attention given to the hazards and probability of failure associated with the various components. For the present analysis, the following provisions are assumed:

1 Power System
   Redundant prime mover and controls (expander and fuel cell)
   Batteries for 12 kwhr output (solar power unit)
2 Attitude Control System
   Redundant high thrust system for 1000 lb sec output
   Redundant low thrust system for 4 hr duration
3 Life Support System
   Potassium superoxide backpack system for 4 hr operation

The following paragraphs are devoted to discussion of the design of the three integrated systems to be studied:

<u>Oxygen-Hydrogen Bipropellant Expander System</u>

Since primary consideration is being given to long-duration missions in this comparison to ascertain the limits of system applicability, the expander will be operated at conditions consistent with low specific propellant consumptions, as follows:

Inlet temperature = 1800 F
Cycle efficiency = 50 per cent
Coolant top temperature = 200 F (Q/P = 0.54)
Specific propellant consumption = 1.12 lb per bhphr

Assuming a generator efficiency of 85 per cent leads to a design propellant flow of 5.29 lb per hr, at an oxygen-hydrogen ratio of 0.90.

An additional heat sink providing for dissipation of 46 per cent of the expander shaft power output (5550 Btu per hr) and for dissipation of the metabolic heat load (1800 Btu per hr)

TABLE 5

ESTIMATED INTEGRATED SYSTEM WEIGHT - HYDROGEN-OXYGEN BIPROPELLANT EXPANDER

	FIXED WEIGHT	VARIABLE WEIGHT
POWER SUPPLY		
Expander Prime Mover	35 lb	- lb per hr
Associated Valves and Controls	37	-
Cryogenic Hydrogen	17	2.77
Cryogenic Oxygen	15	2.52
Hydrogen Tankage	13	0.83
Oxygen Tankage	12	0.84
Batteries		-
THERMAL CONTROL SYSTEM		
Radiator	32	-
Heat Transfer Loop	85	-
Expendable Evaporant (H_2O)	137	-
Evaporant Tankage	19	-
WATER MANAGEMENT SYSTEM		
Water Recovery Unit	57	-
Water Supply + Makeup + Filters	10	0.07
ATTITUDE CONTROL SYSTEM		
Low Thrust, Long Duration	45	-
High Thrust, Short Duration	37	-
ATMOSPHERE CONTROL SYSTEM		
Carbon Dioxide Removal	50	-
Water Vapor Removal	16	-
Oxygen Makeup + Tankage	75	0.36
Nitrogen Makeup + Tankage	92	0.03
Valves, Fans, Controls, Etc.	42	-
FOOD AND SANITATION	22	0.20
STANDBY/EMERGENCY EQUIPMENT		
Expander Prime Mover	35	-
Associated Valves and Controls	29	-
Attitude Control	82	-
Backpack Life Support System	102	-
TOTAL INTEGRATED SYSTEM WEIGHT	1114 lb	7.62 lb per hr

is required. For an average radiator temperature of 40 F, this heat load requires a radiator panel with an area of approximately 90 sq ft for radiation free space. During the boost and reentry phases of operation, water as an expendable evaporant provides cooling.

The expander discharge gas temperature is 670 F which corresponds to a specific impulse of 323 sec for an infinite pressure ratio and a nozzle efficiency of 1.0. Assuming a pressure ratio of 100 and a nozzle efficiency of 85 per cent gives a specific impulse of 257 sec or a thrust of 0.38 lb for the propellant flow of 5.29 lb per hr. By combustion of the discharge gas with additional oxygen, the specific impulse can be increased, it being possible to obtain controllable thrust of the order of 1 lb in this way. If vehicle dynamics require higher peak thrusts, it may be feasible to store low-level energy in the form of flywheels or gas accumulators.

Standby or emergency operation is provided by a redundant power unit and control system. Propellant-system reliability is insured through use of multiple tanks. Batteries provide power for ventilation and operation of the radio beacon after landing for the period prior to recovery of the capsule.

Oxygen-Hydrogen Fuel Cell System

With fuel cells, all of the free energy of reaction (ΔF^o = -5450 Btu per lb) is theoretically available for conversion into electrical output. Actually, depending upon the type of fuel cell and its design operation, 50-70 per cent of the free energy change is converted into useful electrical output. These efficiencies correspond to specific propellant consumptions of 0.67-0.93 lb per hphr. Allowing an efficiency of 85 per cent for the functions of control and conversion of the electrical output in addition to an assumed fuel-cell efficiency of 70 per cent gives a fuel requirement of 3.16 lb per hr for 3 kw output.

Internal heat generation in the fuel cell is given by the difference between the enthalpy of reaction (ΔH^o = -5770 Btu per lb) and the electrical power output. The heat load to be dissipated by the environmental control system consists of the following:

	Btu/hr
Fuel-cell internal heat load	6170
Equipment heat load	12030
Metabolic heat load	1800
Total heat load	20000
Propellant cooling capacity	1310
Net heat rejection to radiator	18690

Sizing the radiator to dissipate this heat load at an average radiator temperature of 40 F leads to a radiator area of 229 sq ft. Water as an expendable evaporant provides cooling during boost and reentry.

Exhaust gas from the fuel cell will consist of essentially pure water vapor at a temperature level depending upon the type of fuel cell used. Assuming a discharge temperature of 200 F, a pressure ratio of 100 and a nozzle efficiency of 85 per cent gives an available thrust of 0.083 lb. This will probably be too low to be of much value in attitude control. The water vapor discharged from a fuel cell can be better utilized by condensation and collection for use in the drinking supply, thereby eliminating need for a water-recovery unit. Assuming condensation at 200 F, this will require an additional radiator surface area of 15.2 sq ft. The waste water is utilized in the attitude control system to provide thrust to minimize the propellant requirements.

TABLE 6
ESTIMATED INTEGRATED SYSTEM WEIGHT - HYDROGEN-OXYGEN FUEL CELL POWER

	FIXED WEIGHT	VARIABLE WEIGHT
POWER SUPPLY		
Fuel Cell	300 lb	- lb per hr
Static Inverter and Regulator	42	-
Associated Valves and Controls	30	-
Cryogenic Hydrogen	3	0.35
Cryogenic Oxygen	19	2.81
Hydrogen Tankage	11	0.21
Oxygen Tankage	13	0.94
THERMAL CONTROL SYSTEM		
Radiator	82	-
Heat Transfer Loop	111	-
Expendable Evaporant (H_2O)	202	-
Evaporant Tankage	25	-
WATER MANAGEMENT SYSTEM		
Water Condenser-Radiator	15	-
Water Supply	9	-
ATTITUDE CONTROL SYSTEM		
Low Thrust, Long Duration	73	1.87
High Thrust, Short Duration	37	-
ATMOSPHERE CONTROL SYSTEM		
Carbon Dioxide Removal	50	-
Water Vapor Removal	16	-
Oxygen Makeup + Tankage	75	0.36
Nitrogen Makeup + Tankage	92	0.03
Valves, Fans, Controls, Etc.	42	-
FOOD AND SANITATION	22	0.20
STANDBY/EMERGENCY EQUIPMENT		
Fuel Cell	300	-
Associated Valves and Controls	30	-
Attitude Control	180	-
Backpack Life Support System	102	-
TOTAL INTEGRATED SYSTEM WEIGHT	1811 lb	6.77 lb per hr

TABLE 7
ESTIMATED INTEGRATED SYSTEM WEIGHT - SOLAR POWER UNIT

	FIXED WEIGHT	VARIABLE WEIGHT
POWER SUPPLY		
Solar Power Unit	750 lb	- lb per hr
Batteries	189	-
Static Inverter	39	-
THERMAL CONTROL SYSTEM		
Radiator	65	-
Heat Transfer Loop	92	-
Expendable Evaporant (H_2O)	154	-
Evaporant Tankage	21	-
WATER MANAGEMENT SYSTEM		
Water Recovery Unit	57	-
Water Supply + Makeup + Filters	10	0.07
ATTITUDE CONTROL SYSTEM		
Low Thrust, Long Duration	73	3.12
High Thrust, Short Duration	37	-
ATMOSPHERE CONTROL SYSTEM		
Carbon Dioxide Removal	50	-
Water Vapor Removal	16	-
Oxygen Makeup + Tankage	75	0.36
Nitrogen Makeup + Tankage	92	0.03
Valves, Fans, Controls, Etc.	42	-
FOOD AND SANITATION	22	0.20
STANDBY/EMERGENCY EQUIPMENT		
Batteries	282	-
Attitude Control	129	-
Backpack Life Support System	102	-
TOTAL INTEGRATED SYSTEM WEIGHT	2291 lb	3.78 lb per hr

Batteries are not required with a fuel-cell system, since fuel cells operate efficiently at low power levels, and are essentially independent of discharge pressure. Emergency operation is provided by a redundant fuel-cell and control system, as with an expander.

Solar-Power-Unit System

In this system, power is provided with a Rankine-cycle heat engine, using a solar concentrator as the energy source. The weight of this system is estimated to be 750 lb, including energy storage for dark-side operation. Batteries provide power prior to erection of the solar collector and during reentry.

The environmental control system dissipates a heat load consisting of 3 kw arising from the equipment and 1800 Btu per hr from the occupants. A radiator sized to dissipate this heat load at an average temperature of 40 F will have an area of 148 sq ft.

Since the power unit produces no waste gases, the attitude control requirements must be supplied entirely from a special system provided for that purpose. The attitude control requirements for this type of system will probably be more critical than those for the other two, because of the accuracy of sum tracking required for proper operation of the solar collector.

Tables 5, 6 and 7 contain weight comparisons for the three types of systems under consideration for the hypothetical manned orbital vehicle. The weight estimate is divided into two quantities: a fixed weight, which is determined by the requirements with respect to boost, reentry and emergency modes of operation; and a variable weight, which is dependent upon the time in orbit. Table 8 summarizes the weight comparison

Table 8 Summary of Weights

	Fixed weight, lb	Variable weight, lb/hr
H_2-O_2 expander . .	1114	7.62
H_2-O_2 fuel cell. .	1811	6.77
Solar power unit .	2291	3.78

In this case, the oxygen-hydrogen expander will be superior to the fuel-cell system and the solar-powered system for durations less than 306 hr. The expander system weights are based upon the assumption that expander exhaust, which provides a maximum thrust of 0.38 lb, can be used for attitude control. Little information is available concerning vehicle attitude control re-

quirements, although some investigators of vehicle dynamics hold that although an average thrust level of this magnitude may be adequate, higher peak thrusts will be required to insure vehicle stability. In that event, the weights relating to the attitude control system can be subtracted from all three systems to give the comparative system weights in Table 9 (assuming use of an exhaust-gas recuperator on the expansion cycle to reduce SPC to 1.05 lb per bhphr):

Table 9 Adjusted Weight Comparison

	Fixed weight, lb	Variable weight, lb/hr
H_2-O_2 expander . .	957	6.92
H_2-O_2 fuel cell. .	1591	4.90
Solar power unit .	2058	0.66

Below 314 hr, the expander is superior to the fuel cell, although the solar system is superior to both for durations greater than 176 hr.

Certain limitations in the foregoing analysis should be indicated:

1 The systems shown do not represent optimum configurations. In addition, the weights given are not necessarily complete in allowing for plumbing, mounting provisions, and so on, but are relative.

2 The tankage weights are quite approximate in representing attainable performance and do not accurately reflect penalties for long-duration storage.

3 The fuel-cell specific weights which were assumed are indicative of the present state of the art and may be substantially reduced.

4 Expander-system performance is based upon simple cycles. Substantial improvements in performance are possible with more complex cycles and with refinement in prime mover design.

CONCLUSIONS

Cryogenic hydrogen can be used in multipurpose systems for spacecraft advantageously for durations from 1 hr to durations substantially in excess of 100 hr. The maximum duration for optimum application depends upon the vehicle attitude control requirements and the ability to utilize the large total impulses, at low thrust levels, contained in the working fluid discharged from an open-cycle expansion power unit.

In short-duration applications (less than approximately 19 hr), the system will be operated with the propellant flow capable of absorbing the vehicle heat load to attain minimum system weight. For longer durations, the system will be designed for the minimum specific fuel consumption, using space radiators to reject the heat in excess of the cooling capacity available in the propellant flow.

In order to make optimum use of a cryogenic-hydrogen system, consideration must be given to energy requirements of the other major systems on the vehicle. The characteristics of cryogenic hydrogen which make it attractive in integrated systems can be summarized as follows:

1 Low molecular weight - high energy working fluid.

2 Very low temperature heat sink.

3 High specific heat - high thermal capaci[illegible] heat sink.

4 High heat of combustion with oxygen.

Thus, hydrogen acts as a high-performance working fluid in a power cycle and as a high-performance propellant for attitude control. The low-temperature cooling capabilities can be used to advantage for such purposes as atmosphere contaminant freezeout and infrared detector cooling.

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